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Thomas W. Kerslake, Jeffrey S. Hojnicki, Robert D. Green, and Jeffrey C. Follo
Lewis Research Center
Cleveland, Ohio

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Thomas W. Kerslake, Jeffrey S. Hojnicki, Robert D. Green, and Jeffrey C. Follo
National Aeronautics and Space Administration
Lewis Research Center
Cleveland, Ohio 44135

One of the most important resources of Space Station **Freedom** is electric power. The Electric Power System (EPS) capability to effectively deliver power to housekeeping and user loads continues to strongly influence **Freedom's** design and planned approaches for assembly and operations. The EPS design consists of silicon photovoltaic (PV) arrays, nickel-hydrogen batteries, and direct current power management and distribution (PMAD) hardware and cabling (Thomas and Hallinan, 1989). A dedicated active thermal control system maintains batteries and electrical equipment within their prescribed operating temperature limits.

To properly characterize the inherent EPS design capability, detailed system performance analyses must be performed for early assembly stages as well as for the fully assembled station up to 15 years after beginning of life. Such analyses have been repeatedly performed using the FORTRAN code SPACE (Station Power Analysis for Capability Evaluation) developed at the NASA Lewis Research Center over a 10-year period. SPACE combines orbital mechanics routines, station orientation/pointing routines, PV array and battery performance models, and a distribution system load-flow analysis to predict EPS performance. Time-dependent, performance degradation, low earth orbit environmental interactions, and EPS architecture build-up are incorporated in SPACE. A companion paper describes SPACE (Hojnicki, et al., 1993).
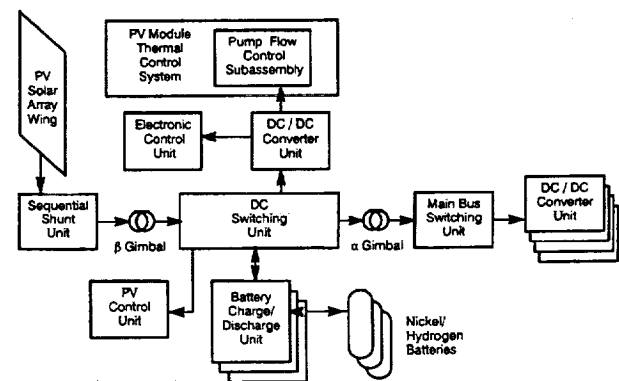
In this paper, results from two typical SPACE analytical cases are presented: (1) an electric load driven case and (2) a maximum EPS capability case. In the former, EPS ability to meet a specified load profile during Space Shuttle Orbiter rendezvous, berthing and post-berthing with the fourth mission build hardware

(MB-04) is assessed. In the latter, the maximum continuous day/night EPS capability under nominal orbital conditions after 5 years of station operation is predicted. For each analysis case, inputs are summarized and the results are discussed with respect to EPS requirements and ground rules.

Analysis Inputs

Load Driven Case

The MB-04 mission is planned for July 1996. **Freedom** is configured with one starboard Photovoltaic Module (PVM). A block diagram of the PVM and PMAD hardware is shown in Fig. 1. The PVM has two solar array

Fig. 1. EPS Architecture Block Diagram

wings that power two electrical distribution channels designated #1 and #3. Each channel has an energy storage subsystem with 1 out of 3 batteries not included or *off-loaded* (implemented due to launch constraints). The primary bus provides input power to a total of 7 DC-to-DC Converter Units (DDCUs). At the DDCU output, referred to as *Interface A*, power is supplied to the secondary power distribution system.

During the 140 orbit MB-04 mission, **Freedom** operates in a circular 220 nmi altitude orbit with a 33° to 52° solar beta angle (angle between orbit plane and plane of the ecliptic) and solar insolation of 1330 W/m^2. Two-axis solar array articulation is provided by alpha and beta gimbals to track the orbital and seasonal sun position, respectively. Rendezvous and berthing events occur between orbits #30 and #35 (McCormick and Workman, 1993) with **Freedom** in gravity gradient (with **Freedom's** truss along the local vertical and the solar array wings positioned away from earth) or free drift flight attitudes. The solar array wings lie in the orbit plane, permitting only beta angle sun tracking, for orbits 30 - 32, are locked for orbits 33 - 35 and lie out of the orbit plane, permitting only alpha angle sun tracking, for orbits 36 and after. For each orbit, the solar array operating voltage is set at a constant 8 volts lower than the smallest array maximum power point voltage, V_{mp}. V_{mp} is generally smallest at orbital noon when array temperatures are highest.

The Interface A electrical load demand for channels 1 and 3 was generated by NASA Johnson Space Center Code ET and is shown in Fig. 2. The starting point for each orbit is the beginning of eclipse and the orbit number position on the abscissa represents the end of the particular orbit. Individual DDCU load levels vary between 0.2 and 1.0 kW with a total load level between 3 and 3.5 kW.
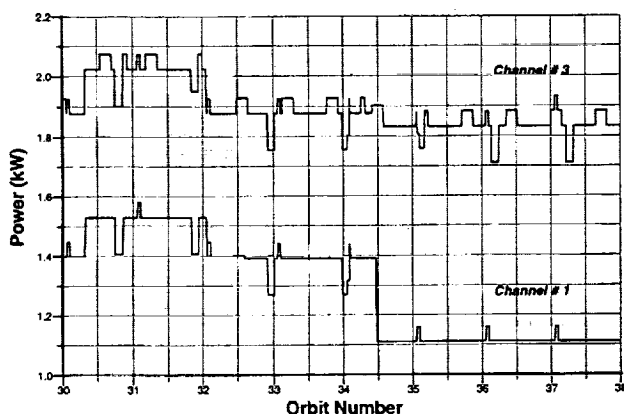

Fig. 2. Interface A Power Demand

Maximum Continuous EPS Capability Case

For this analysis, it is assumed that all **Freedom** hardware has been launched and operated on orbit for 5 years (beginning-of-life plus 5 years, i.e. BOL+5). The EPS consists of 4 PVMs with the associated PMAD hardware and 26 evenly loaded DDCUs. The EPS architecture is the same as that in Fig 1. with the exception that PVM channels #1 and #3 both have full sets of 3 batteries. Solar array and battery performance is degraded consistent with 5 years of orbital operation. See the companion paper for degradation modeling (Hojnicki et al., 1993).

Freedom operates in a 200 nmi, circular orbit with a seasonal average solar beta angle of 27° and a minimum solar insolation of 1326 W/m^2. **Freedom's** flight mode is local-vertical, local horizontal with full solar array alpha and beta angle tracking.

Through iterative calculations, the maximum EPS interface A power that is sustainable through insolation and eclipse periods is determined. For these calculations, the solar array operating voltage is set to the minimum V_{mp} minus 8 volts and the batteries are limited to a 34% depth-of-discharge (DOD) to permit a 5 year operating life. The batteries are charged at essentially constant current up to a 96% state of charge (SOC). At this point, the charge current is tapered in a time-linear fashion to 2 amps (A). This is done to improve the battery operating lifetime and to reduce excessive heat generation as the batteries approach a state of full charge.

Results

Load Driven Case

The power demand in Fig. 2 can be supplied by the EPS. Hence, no load shedding is required. Fig. 3 shows the available solar array wing power and the number of active (unshunted) solar array wing circuits (or strings) out of a total of 82. For orbits 30 - 32 without

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alpha angle pointing, the variation in available wing power roughly follows the cosine of the angle between the sun vector and the array surface normal. This angle is smallest at orbit noon resulting in the highest array power. Deviations from cosine behavior result from two effects: (1) angle-dependent solar cell cover glass reflectivity and (2) temperature-dependent solar cell power influenced by the orbital variation in array environmental heat loads and the concomitant variation in array temperature. Also note that no array power is produced in the early and late portions of the orbit sun period when the array edge or backside is sun facing. This effectively increases the orbital eclipse time which requires greater battery energy storage.
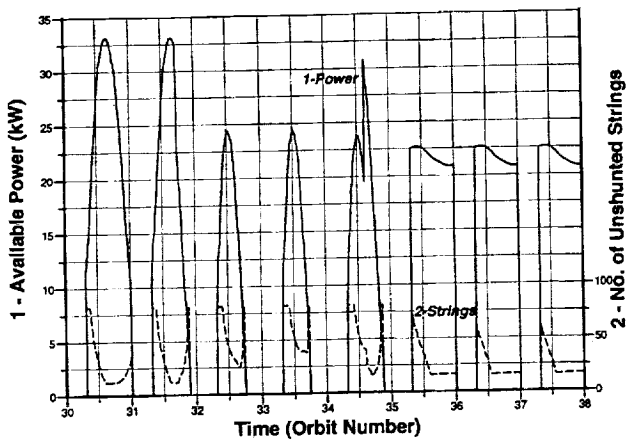

Fig. 3. Available Wing Power and Number of Active Strings

During orbits 33 - 35, the alpha and beta gimbals are both locked for the Orbiter approach and berthing. Sun tracking efficiency is further reduced over the previous orbits since a beta angle error is also introduced. This reduces the maximum array power available at orbit noon in addition to reducing the fraction of orbit sun time that the array produces power. Midway through orbit 35, **Freedom's** flight attitude changes to produce a more favorable array pointing situation and a localized peak in array power.

For orbits 36 and after, the arrays are positioned out of the orbit plane allowing orbital (alpha) sun tracking via the beta gimbal. This

results in a relatively flat orbital array power profile. The 1 kW to 2 kW power variation is the result of small torque-equilibrium angular offsets in **Freedom's** attitude. The array power level is consistent with the solar beta angle error which exists for this particular **Freedom** flight mode.

For all orbits, the sequential shunt unit (SSU) responds to variations in available array power by shunting fewer array strings to make-up for the power loss from off-pointing or shunting more strings when array power levels are high. This allows the SSU to deliver the power necessary for the EPS to meet the sun light load demand and fully charge the batteries.

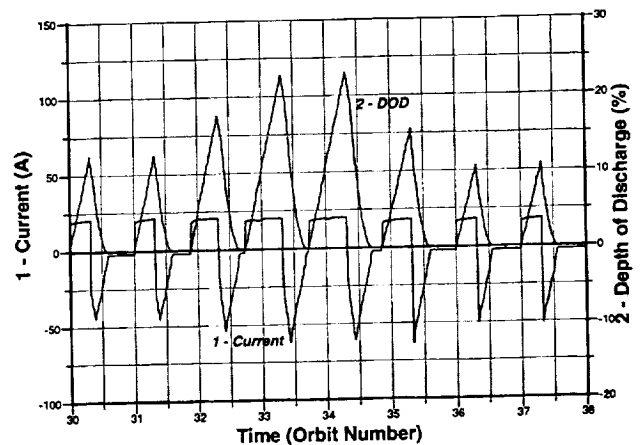Fig. 4 shows the battery DOD and charge-discharge current. By convention, a negative

Fig. 4. Battery DOD and Current Profiles

battery current is associated with battery charging. The maximum DOD of 23% occurs on orbits 34 and 35. For these orbits, large effective eclipse times occur (due to array off-pointing) as well as non-zero initial DOD values (i.e., 10%). The 23% DOD value is well below the 34% DOD value recommended to permit 5 year battery life.

The maximum allowable battery charge current is 85 A but was set to 65 A for this analysis. For orbits 34 - 36, the 65 A limit is reached prior to the start of the taper charge at a 96% SOC. The lower current limit was selected to reduce battery wear and to moderate battery heat generation. Since the charge current limit can be set during actual EPS orbital

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operation, an optimum value could be selected to meet the anticipate load demand (with margin for power peaking capability) while minimizing battery wear.

Fig. 5. shows various EPS component heat dissipation rates. The battery heat generation rate profile has two distinctive peaks for orbits 30 - 35. The lower peak of 0.3 kW occurs at the end of the discharge period. Note that the discharge period starts during the sun light portion of the preceding orbit due to low solar array power from poor pointing conditions. The higher peak at 0.65 kW corresponds to the high charge current period just prior to the taper charge period. The heat generation rate then decreases sharply in response to the falling current levels during taper charge.

The battery charge-discharge unit (BCDU) heat generation profile has peaks each orbit at values ranging from 0.45 kW to 0.6 kW. This peak is coincident with the high charge current at the start of battery charging. For the remainder of the orbit, the heat generation rate is constant at 0.35 kW during charge mode operation and 0.38 kW during discharge mode operation. The DC switching unit (DCSU) and DDCU heat rates, which are summed together, also exhibit a small peak at the start of battery charging. This peak is associated with high DCSU current levels that exist to supply the BCDU power for battery charging.
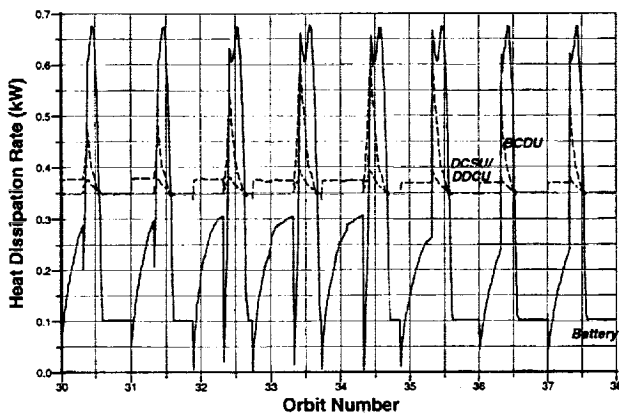

Fig. 5. Component Heat Dissipation Rates

Maximum EPS Capability Case

At BOL+5, the maximum continuous EPS capability per PVM at interface A is 20.4 kW while the insolation EPS capability is 23.5 kW. This provides a 9% margin over the 18.75 kW requirement to account for analytical uncertainties and future performance modeling updates. These results indicate that EPS performance is in a *battery limited* condition, i.e. the maximum DOD limit of 34% has been reached and the sunlight EPS capability exceeds the eclipse capability.

Fig. 6 shows several solar array wing performance parameters through the analyzed orbit. Orbit time zero corresponds with the start
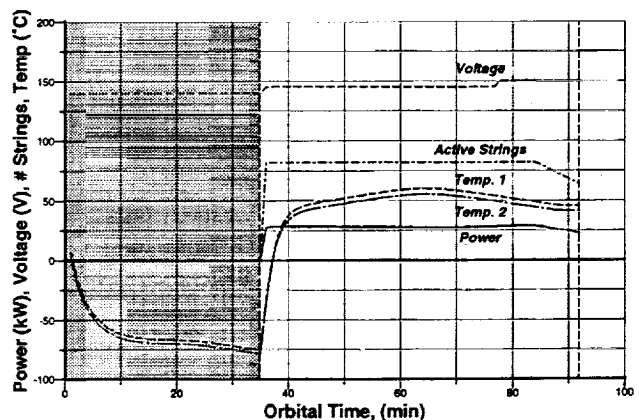

Fig. 6. Wing Performance Parameters

of eclipse. Wing power is nearly constant at 28 kW for most of the insolation period but decreases slightly near orbital noon when the highest wing temperatures occur. For the last 9 minutes of the orbit, wing power decreases proportionally with the number of active strings attaining a minimum value of 22.7 kW. This is due to the battery taper charge period which requires less array power.

During eclipse, the wing voltage, 140 V, is approximately 1 volt less than the channel voltage which is controlled by the BCDU. During the first portion of insolation, the BCDU continues to control channel voltage by regulating the battery charge current. During

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this period, the wing operating voltage is about 145 V. Once the BCDU starts the battery taper charge period at 77 minutes, the SSU controls the channel voltage (by shunting or unshunting strings and by pulse-width modulation switching of a single string at 20 kHz). For this period, the wing voltage is about 150 V since the SSU voltage set point is about 5 V higher than the BCDU set point. At this operating voltage, wing power increases by 700 W.

All 82 array wing strings are active (unshunted) during insolation until 83 minutes. After 83 minutes, the battery charge current decreases enough so that full array power is no longer needed. With the interface A load demand constant, the SSU shunts array strings in proportion to the fall off in charge current.

The two solar array blankets that comprise each wing operate at temperatures within 3°C to 5°C of each other. The minimum blanket temperature, -78°C, is attained at the end of eclipse. As **Freedom** enters the sunlit portion of the orbit, the blankets attain a pseudo-steady temperature of 40°C in about 5 minutes. An orbit noon maximum temperature of 60°C is reached as the heat load from Earth emitted/reflected energy is highest. Blanket 1 temperatures (Temp. 1 in Fig. 6) are greater than those of blanket 2 (Temp. 2) due to a higher blanket-to-blanket view factor between adjacent PVMs, i.e. outboard blanket 1 on the inboard PVM has a good view to inboard blanket 2 on the neighboring outboard PVM. At a beta gimbal position of 0°, all the blankets of companion PVMs are essentially coplanar and the blanket-to-blanket view factor is 0. In this situation, blankets 1 and 2 would operate at the same temperature.

Fig. 7 shows battery performance parameters throughout the orbit. Over the entire orbit, the predicted battery round trip (watt-hour) efficiency is 80%. A constant battery discharge power of 4.5 kW is required through eclipse to meet a constant power demand. Since battery operating voltage falls from 108 V to 90 V over the discharge period, discharge current levels increase from 43 A to 51 A to compensate. Battery heat dissipation increases sharply to a
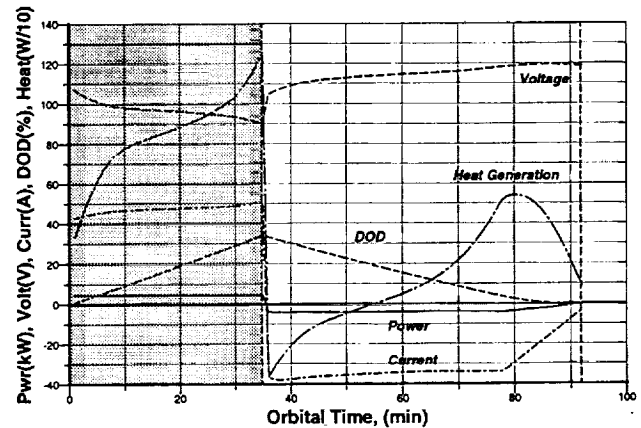

Fig. 7. Battery Performance Parameters

peak of 1.25 kW at the end of discharge.

During insolation, the battery is charged at a nearly constant 4 kW power level up to 77 minutes when the taper charge period starts. Prior to the taper charge, the BCDU regulates battery charge voltage to provide the proper charge current. The available charge current decreases slightly toward orbit noon. The charge current is reduced to permit a constant interface A power level while the solar array output power is diminished due to high noon time temperatures.

During charging, the batteries initially operate endothermically and thus, absorb heat at a rate up to 0.36 kW instead of liberating heat. The batteries then operate exothermically dissipating up to 0.54 kW just prior to the taper charge period. Without a taper charge period, the heat dissipation rate would have continued to increase until the end of the charge period.

Concluding Remarks

Freedom EPS performance predictions were presented for two important types of analyses: (1) a load driven analysis and (2) a maximum continuous capability analysis. The former analysis type is useful for planning mission electrical load schedules and for assessing the EPS performance impacts of planned **Freedom** flight modes. The latter

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analysis type is useful for long term (month-to-month and year-to-year) scheduling of loads and for planning **Freedom** hardware maintenance missions such as replacing batteries. The results presented demonstrate the capability of the FORTRAN code SPACE to model detailed EPS performance under a broad range of conditions.

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Hojnicki, J. S., et al., "Space Station Freedom Electrical Performance Model," <u>28th Intersociety Energy Conversion Engineering Conference Proceedings</u>, (in press), Atlanta, Georgia, August 8-13, 1993.

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13. ABSTRACT (Maximum 200 words)

Space Station Freedom Electric Power System (EPS) capability to effectively deliver power to housekeeping and user loads continues to strongly influence Freedom's design and planned approaches for assembly and operations. The EPS design consists of silicon photovoltaic (PV) arrays, nickel-hydrogen batteries, and direct current power management and distribution hardware and cabling. To properly characterize the inherent EPS design capability, detailed system performance analyses must be performed for early assembly stages as well as for the fully assembled station up to 15 years after beginning of life. Such analyses have been repeatedly performed using the FORTRAN code SPACE (Station Power Analysis for Capability Evaluation) developed at the NASA Lewis Research Center over a 10-year period. SPACE combines orbital mechanics routines, station orientation/pointing routines, PV array and battery performance models, and a distribution system load-flow analysis to predict EPS performance. Time-dependent, performance degradation, low earth orbit environmental interactions, and EPS architecture build-up are incorporated in SPACE. In this paper, results from two typical SPACE analytical cases are presented: (1) an electric load driven case and (2) a maximum EPS capability case.

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